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DESIGN STUDY OF A LARGE UNCONVENTIONAL LIQUID PROPELLANT ROCKET ENGINE AND VEHICLE

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(The Martin-Marietta Corporation)

Contract NAS 5-1025

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DESIGN CONSIDERATIONS FOR LARGE BOOSTERS

September 1961

Prepared by

THE ADVANCED MISSILE AND BOOSTER DEPARTMENT

Advanced Programs Division

Thomas R. Heath for

Heinz Mueller, Chief
Advanced Propulsion Section

THE MARTIN COMPANY

Denver 1, Colorado

INTRODUCTION

During the period of June 16, 1961, to August 31, 1961, The Advanced Propulsion Section of the Advanced Missile and Booster Department, The Martin Company-Denver Division has been furnishing consultation and advice to The Aerojet-General Corporation, Liquid Rocket Propellant Plant with regard to the NASA study contract NAS 5-1025. This contract is mainly concerned with the study of rocket engines for large boosters (thrust level of 2,000,000 lb_f to 24,000,000 lb_f), but the effects of the airframe design must also be considered in such a study. It was in this capacity that The Martin Company-Denver Division furnished consultation.

This report is written for the purpose of documenting the more important information which disseminated to The Aerojet-General Corporation in connection with this study.

I. PERFORMANCE ANALYSIS

A sizing and propellant optimization study was conducted for several different one and two stage vehicles to determine the maximum payload that could be placed into a 300 nautical mile orbit. The sizing study for the first vehicle considered was based on the parametric studies conducted by the Martin-Denver Division. The results of the parametric data were compared to IBM trajectory calculations. Because the parametric results agreed with more accurate IBM data, all other performance calculations were based on the parametric study.

An east launch from the Atlantic Missile Range was assumed for all vehicles. The trajectory shape chosen to accomplish the mission was to have a horizontal burnout at 50 nautical miles with sufficient velocity to enter a Hohmann transfer orbit and coast to 300 nautical miles. At 300 nautical miles the additional velocity to reach circular velocity was added.

The first vehicle considered the two stage liquid propellant using liquid oxygen and liquid hydrogen in both stages. A sea level thrust of 2,032,400 was applied and an initial thrust to weight ratio of 1.25 was assumed. The propellant weight and mass fraction in each stage and the second stage thrust was allowed to vary to determine the maximum payload in orbit. The results of the sizing study for this standard booster are as follows:

Liftoff weight	1,625,290 lb_m
1st stage propellant weight	1,066,000 lb_m
1st stage thrust	2,032,400 lb_f
2nd stage initial weight	476,300 lb_m
2nd stage propellant weight	341,340 lb_m
2nd stage thrust	857,340 lb_m
Payload weight in orbit	121,430 lb_m
1st stage theoretical velocity	14,550 ft/sec
2nd stage theoretical velocity	17,270 ft/sec

Two different single stage liquid oxygen liquid hydrogen vehicles were studied to determine their payload capability. The first of these used a conventional bell nozzle rocket engine and the second a plug nozzle rocket engine. For both of these vehicles a total theoretical velocity of 30,350 ft/sec was assumed for the 300 nautical mile mission. The payload capability and characteristics of these two vehicles are as follows:

	Bell Nozzle	Plug Nozzle
Lift off weight, lbs_m	1,600,000	1,600,000
Sea level thrust, lbs_f	2,000,000	2,000,000
Propellant weight, lbs_m	1,438,530	1,399,400
Vacuum specific impulse, $\frac{lbf\text{-}sec}{lb_m}$	411	454
Payload weight (λ* = .9405 & .939)	70,460	109,690
Payload weight (λ = .942 & .942)	72,900	114,440
Payload weight (λ = .9455 & .9455)	78,550	119,940

* = propellant mass fraction

In addition to the standard booster two other stage vehicles were sized and the maximum payload capability for a 300 nautical mile circular orbit determined. One of these vehicles used liquid oxygen and RP_1 in both stages. The other used liquid oxygen and kerosene in the first stage and liquid oxygen and liquid hydrogen in the second stage. The characteristics and payload capability of these two vehicles is as follows:

Propellant		
First stage	Oxygen/Kerosene	Oxygen/Kerosene
Second stage	Oxygen/Hydrogen	Oxygen/Hydrogen
Liftoff weight, lb_m	1,600,000	1,600,000
Sea level thrust, lb_f	2,000,000	2,000,000
1st stage propellant weight, lb_m	1,238,000	1,030,000
1st stage vacuum specific impulse $\frac{lb_f\text{-sec}}{lb_m}$	309	309
2nd stage initial weight, lb_m	299,580	514,650
2nd stage thrust, lb_f	438,000	672,000
2nd stage propellant weight, lb_m	227,800	391,000
2nd stage vacuum specific impulse $\frac{lb_f\text{-sec}}{lb_m}$	322	426
Payload weight in orbit, lb_m	61,800	93,570
1st stage theoretical velocity, ft/sec	14,780	10,280
2nd stage theoretical velocity, ft/sec	14,810	19,660
1st stage propellant mass fraction	.952	.949
2nd stage propellant mass fraction	.955	.931

The basic 2,000,000 lb thrust liquid oxygen liquid hydrogen unit was applied in clustered form to obtain two new vehicles. The first vehicle considered three of the basic units as a first stage and one unit for a second stage. The second vehicle used four of the basic units as a first stage and one unit for a second stage. The characteristics and payload capability of these two vehicles is as follows:

Liftoff weight, lb	4,800,000	6,400,000
Sea level thrust, lb	6,000,000	8,000,000
1st stage propellant weight, lb	3,100,000	4,179,850
1st stage vacuum impulse, $\frac{lb_f\text{-sec}}{lb_m}$	411	411
2nd stage initial weight, lb	1,811,800	1,811,800
2nd stage thrust, lb	2,390,000	2,390,000
2nd stage propellant weight, lb	974,000	1,233,930
2nd stage vacuum specific impulse, $\frac{lb_f\text{-sec}}{lb_m}$	426	426
Payload weight in orbit, lb	377,430	487,840
1st stage theoretical velocity, ft/sec	13,740	14,000
2nd stage theoretical velocity, ft/sec	15,840	15,663
1st stage propellant mass fraction	.918	.911
2nd stage propellant mass fraction	.932	.932

II. PROPULSION SYSTEM

PROPULSION SYSTEM LO_2 - LH_2 VEHICLE

A. STANDARD OXYGEN - HYDROGEN VEHICLE

1. Propellant Feed System

Insulation. Insulation is effective in reducing the losses caused by the vaporization of cryogenic propellants before the booster is launched and insulation also reduces the heat transfer to the propellants during the flight, which in turn reduces the propellant tank gas pressure needed for proper net head pressure at the engine turbopump. Insulation is especially essential for a liquid hydrogen tank in order to achieve an efficient design. This exists because of the lower amount of heat required to vaporize a unit volume of hydrogen when compared to oxygen. Vacuum, cork, and styrene foams are several insulators for propellant containers. A potential problem exists in maintaining a high vacuum in the presence of a vibration environment.

Since propellant storage facilities are expected to be located a considerably long distance from the launch vehicle, insulation of the propellant transfer lines can be highly useful, if not necessary, in reducing the vaporization of the propellant during propellant transfer.

Baffles, Slosh Suppressors, Propellant Entrances. Baffles are required to prevent vortexing or propellants generated at the entrances of the propellant feed lines that, subsequently, enables full utilization of the propellants. Slosh suppressors dampen the oscillation of the propellants thereby allowing more stable control of the vehicle. An effective method for reducing propellant oscillation is with baffles located circumferentially within the propellant tanks. The entrances for the propellant feed lines should be shaped to eliminate cavitation and reduce the pressure loss of the system in order that the gas pressure can be held to a minimum, thereby required by the gas system and the propellant tank weights. A typical entrance profile is given in figure 1 and is found by the following equation: $h = \left(\frac{Q}{D^2}\right)^2/2g$

where h = fluid pressure above vapor pressure, ft.

Q = fluid volumetric flow rate, ft^3/sec.

D = inlet diameter, ft.

g = vehicle acceleration, ft/sec^2.

2. Propellant Utilization

Both a propellant utilization subsystem and a calibrated engine are two methods for controlling the residual propellant existing at the end of flight. The calibrated engine method will produce residual propellant amounting to approximately one percent of the total propellant weight. A propellant utilization subsystem will produce approximately 0.2% residual propellant of the total initial propellant load. The application of a propellant utilization subsystem can be expected to lower the reliability of the propulsion system.

3. Pressurization System

Propellant pressurization systems provide additional propellant pressure at the pump inlet than can be supplied by the propellant head alone, prevents collapsing of the propellants tanks, and also can increase the payload of the stage by prestressing the loadcarrying members in tension. For the LO_2 - LH_2 stage, helium gas and propellant vapor generating systems, or a combination of the two systems are suitable. Heated helium as the oxidizer pressurant and gaseous hydrogen as the fuel pressurant should produce the least system weight.

The propellant tank location of one with respect to another affects the gas pressure needed to supply the proper head pressure at the pump inlet as given in Figures 2, 3, 4, and 5.

B. STANDARD LIQUID OXYGEN-KEROSENE VEHICLE

1. Propellant Feed System

The location of the propellant tanks with respect to one another is found by the consideration of the tank gas pressurant requirements, the structural weight, the higher temperature oxidizer in the suction line prior to propellant flow, and vehicle stability and control. A layer of higher temperature oxidizer is produced in flight by aerodynamic heating effect on the propellant.

2. Pressurization System

Two types of pressurization systems have been considered: (1) helium to pressurize both propellant tanks; and (2) helium or nitrogen to pressurize the fuel tank and gaseous oxygen to pressurize the oxidizer tank. For a helium system, the helium is stored in spheres within the oxidizer tank, flows through a heat exchanger and enters the propellants tank at 300°F. The second system conserves helium and provides a constant pressure source for pressurizing the oxidizer tank. A helium system can produce the least weight of the two types of systems.

C. RECOVERY CONSIDERATIONS

Recovery of the booster can be essential from a consideration of the frequency of the booster flights. The booster system should be designed for high operating life in order to achieve high reliability and to reduce the maintenance in an attempt to compensate for the loss in booster payload as a result of the recovery system. Recovery systems can exist of simple parachutes to methods of returning the vehicle, self propelled, to the launch facility. A reference is made to a study conducted by The Martin Company on the recovery of the lower booster for a two-stage booster with a launch weight of 1,200,000 lb_m; nitrogen tetroxide, hydrazine plus additive propellants, and a recovery system consisting of lift surfaces and two turbofan engines with 32,000 pounds total sea level thrust, and a cruising range of 250 miles showed the total weight of the recovery system was 49,300 pounds. The total weight of the lower booster at thrust termination of the rocket engines was 87,700 pounds.

(Reference: Martin Report, "Recommended Recoverable Booster System," dated 2 November 1960.)

III. STRUCTURE

A. STANDARD OXYGEN-HYDROGEN VEHICLE

1. The first consideration involves the propellant tanks. The propellant tank barrel section design falls into four main categories:

a) Monocoque;

b) Pressure stabilized;

c) Waffle stiffened;

d) Stringer and frame stiffened.

Pressure stabilized tanks (balloon tanks) withstand all handling and prelaunch loads by means of internal pressurization. Although there is usually a small weight advantage associated by use of pressure stabilized tanks, it can be offset by the fragility of the vehicle. Leaks or punctures which might occur during assembly and ground handling will cause catastrophic collapse of the tank. Monocoque tanks as discussed here are those whose barrel skin is capable of withstanding all ground handling and prelaunch loads without stiffeners and without internal pressure. Unless the tank pressure is so high that it requires a skin with sufficient thickness to carry these loads, a large weight penalty will occur by use of a monocoque barrel design. In the case where prelaunch loads (or other ground loads) exceed the monocoque load-carrying ability of the barrel skin (as designed by pressure) by a small margin, waffle stiffened skin can be applied to provide additional strength. The waffle pockets are machined or chem-milled out of the sheet leaving lands which stiffen the barrel. A waffle design is required when the amount of load which cannot be carried by the skin alone is so small that the stringers and frames required to carry this load would be diminutive and impractical to fabricate. When the loads imposed on the unpressurized tank barrel are much in excess of the load-carrying ability of the skin, stringers and frames will most efficiently supply the additional strength. This type of design is usually required in lower stage tanks.

Tank domes are commonly of four basic shapes:

a) Hemispherical;

b) Ellipsoidal;

c) Segment of sphere;

d) Conical.

Hemispherical domes are the lightest for the volume contained, but often the complete stage will be shorter and somewhat lighter if the tanks are closed by ellipsoidal or segment-of-sphere domes. In metal ellipsoidal domes the highest ratio of major-to-minor axis commonly used is $\sqrt{2}$. Higher ratios can be used without imposing excessive hoop compression on the dome if all proportions of the dome and barrel are chosen correctly. Conical domes are only advantageous where they are also used to transmit engine thrust loads to the shell of the vehicle. At the conical dome-barrel juncture a toroidal segment (rounded) intersection may be used if the dome will be stabilized by sufficient pressure during engine thrust. Dome stringers are not usually required with such a design. If sufficient pressure is not available during engine thrust, the cone can be stiffened by frames and stringers and a heavy frame used at the dome-barrel juncture (not a toroidal intersection) to react the dome and barrel discontinuity loads.

2. Engine Mounts

One means of attaching the engine is by use of a truss, which supports the engine from the outer shell, has the advantage of also supplying structure for attaching pressure bottles, control units, etc. Another means is a cone, which also acts as a tank dome, is sometimes a lighter design since it acts in a dual load-carrying capacity. A third major type of mounting arrangement is the integral tank-engine design. In such a design the engine bell is also used as a reversed tank dome. Both engine thrust and tank dome pressure loads are reacted directly at the barrel. The weight of this design is dependent on tank pressure as compared to combustion pressures on the engine bell, length savings in the tank skirt, and weight required for thrust deflection to substitute for engine gimbaling. Difficulties will be encountered in hydrostatic testing of the tank when the engine is inoperative, in routing the propellant to the engine during the last few seconds of burning, and in engine inspection and replacement.

3. Weights Analysis

The mass fraction curve, Figure 6, is a result of a parametric study to provide a rapid estimate of inert weights. This curve is based on average values for most factors and does not reflect a detail analysis for any configuration. Structural components were computed on the basis of rigid construction, semi-monocoque aluminum skin stringer structure, in both the tank and non-tank areas. The tanks were computed with a gas pressure of 35 lb_f/in^2 in the separate tanks and insulation was not included for any propellant combinations. It should be pointed out that a thickness of 0.125 inches of cork is sufficient to negate the effects of aerodynamic heating on the hydrogen tank. Domes were elliptical (axes ratio = $\sqrt{2}$) except the aft

IV. AERODYNAMICS

A. STANDARD OXYGEN-HYDROGEN VEHICLE

The following Aerodynamic considerations are to be studied in booster design.

Aero heating Effects

Studies of hydrogen-fueled ballistic missiles have shown that aerodynamic heating has little, if any, influence on structural or configuration design. The reason for this is that tank insulation is necessary for per-launch "holds", approaching twelve hours in some cases which precludes that required by the flight environment. Of the various insulations available, the most appropriate appears to be either cork or vacuum jacket. As the cork thickness is reduced from one inch the structural temperatures will show increases but these temperatures are tolerable down to cork layer depths of 0.125 inch. These comments concerning the hydrogen tank are also applicable to external hydrogen and lines.

The extreme boat-tail with the exposed engine nozzle indicates a possible heating problem on the nozzle. No further comments can be made concerning this possible problem because this configuration type has never been studied.

Design Consideration

Aerodynamic design factors are difficult, if not impossible, to evaluate because of a lack of knowledge of the upper stage. In addition the extreme boat-tail may require special attention from a stability and control aspects. It appears, however, that the boat-tail is so extreme that it may be equivalent to a blunt base due to separated flow. The nozzle heating problem mentioned previously may still exist even with separated flow.

If a winged payload is carried the booster will probably require fins for stability.

B. AERODYNAMICS, STANDARD OXYGEN-KEROSENE VEHICLE

The following are Aerodynamic considerations for the Oxygen-kerosene vehicle.

Aerodynamic Heating Effects

Usually the oxygen tanks does not essentially require insulation for long per-launch "holds" and for in-flight propellant heating, the structure is more susceptible to flight environments than the oxygen-hydrogen vehicle. However, the structure required for acceleration and airloads is usually adequate from an aeroheating consideration. The structural temperatures are dependent upon trajectories of the missile.

Design Consideration

Same applies as with the oxygen-hydrogen vehicle.

V. BOOSTER CONTROL SYSTEM

The primary stability problem associated with liquid boosters of large size is the adverse effect of structural bending modes on the low frequency airframe-autopilot dynamics. Proximity of the first structural mode to the closed loop system frequency (approximately 1.0 cycle per second and 0.3 cycles per second respectively) precludes the use of standard filtering techniques. Instead, it is necessary to consider more refined hardware techniques to reduce or eliminate the structural feedback signal in the control loop. Two current approaches to the problem are (1) signal reduction by tuning a quadratic filter to the bending mode frequency and (2) signal cancellation by comparing high frequency output of two gyros located at separate missile body stations.

Control requirements for the first stage are defined primarily by the winds experienced in the region of maximum dynamic pressure. Current design criteria will produce angles of attack up to 10 degrees in a relatively short time. Therefore, in addition to maintaining an adequate static control balance, the vehicle control must respond rapidly enough to prevent large dynamic overshoots. Significant reduction of peak angle of attack can be accomplished by use of auxiliary feedback loops which command the engine gimbal angle as a function of the angle of attack or normal acceleration. Aerodynamic fins will also reduce the angle but have the disadvantage of adding weight and structural complexity to the system. It should be noted that a previous Air Force requirement indicated that fins would be necessary to maintain certain control fixed stability margine on boosters used for manned flight. This approach has been modified however so that a booster without fins can meet the man rating requirements.

Conventional hydraulic actuators with their inherent high reliability and relatively low weight appear to offer the best approach to thrust vector control on liquid propellant systems. Past experience on the use of secondary injection for control of solid boosters has led to the conclusion that this technique increases severe weight penalties due to the necessity of carrying large amounts of control fluid which contributes to the inert weight of the vehicle.

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VI. BOOSTER DYNAMICS

To assure the structural integrity of the missile, its dynamic characteristic during the boost flight must be known. Basically, there are two major areas that must be considered in connection with the design of large boosters: (1) vibration characteristics, including the dynamic stability of the missile along the trajectory, and (2) dynamic loads incurring at launch or in flight through the atmosphere.

A. VIBRATION CHARACTERISTICS

As the missile becomes larger, its natural frequencies will tend to be lower. The fundamental bending frequency of a 1 to 2 million lb_f booster will be generally in the vicinity of 1 cycle per second or lower. Propellant sloshing acts as additional degrees of freedom and couples with the bending modes. Slosh frequencies for large boosters may be in the region between 0.3 and 0.5 cycles per second for the prelaunch condition. They will increase as the acceleration increases and as a consequence of the coupling of slosh degrees of freedom with the bending modes, the bending frequencies increase slightly, perhaps 10%. The detrimental effect of slosh on the stability parameters stems from the fact, that the amplitude of response between the slosh and bending frequencies is higher due to proximity of the two peaks and therefore the stability margin decreases. To compensate for this, some baffles may be required to increase the damping of slosh modes.*

The bending frequencies below approximately 0.8 cycles per second are extremely difficult to handle with conventional autopilots. More sophisticated

* NASA TN D-715, An Experimental Investigation of the Damping of Liquid Oscillations in Cylindrical Tanks with Various Baffles.

approaches such as self-adaptive systems may be required to insure the stability.

Another difficulty lies in the accuracy of calculated freq·.encies. A deviation of ± 10% may easily occur and the experimental verification is difficult, if not impossible.

Considering the clustered boosters, the things are inherently more difficult. In general, the following statements can be applied to clustered boosters: (1) vibration analysis is difficult and inaccurate, (2) frequencies are lower than those for an equivalent integral design, (3) torsional frequencies are extremely low which will increase the complexity of the autopilot.

B. DYNAMIC LOADS

Most severe dynamic loads occur usually at launch. Longitudinal loads are induced due to thrust buildup and due to the release shock. Lateral bending moments occur due to a variety of phenomena such as wind forces, stand and engine misalignment and thrust differential - when more than one engine is used. Of these, the wind forces are predominant and may be divided into static wind forces and random oscillatory forces due to turbulance created at super critical Reynold's numbers by the missile being exposed to a steady wind. Due to these forces the missile executes oscillations in its natural modes (predominately fundamental) in the plane perpendicular to the direction of the wind. The shape of the nose cone plays an important role in the susceptibility to this type of instability.

The gust loads which occur mostly around maximum q condition may contribute 5-10% to bending moments experienced in flight. For lifting body or winged payloads this contribution may be considerably higher.

domes in each stage were conical to provide for thrust loads. Ultimate safety factor for all structure was 1.25.

A conventional type turbo-pump and bell nozzle thrust chamber based on a thrust-to-missile-weight ratio of approximately 1.3 is assumed. Pressure was assumed to be supplied by a gas system which closely approximates the weights of an autogenous type system. Trapped propellants in the system (except the engine) is one tenth of a percent of usable propellant and residual usable propellant at engine shutdown was considered to be two tenths of a percent of usable propellant which is achieved by use of a propellant utilization subsystem. Items which have no growth factor value such as electrical, guidance, destruct systems, etc, were considered constant for all configurations.

B. STANDARD OXYGEN-KEROSENE VEHICLE

1. <u>Propellant Tanks</u>

The same comments apply to the design of tanks for these propellants as were made for oxygen-hydrogen vehicle.

2. <u>Engine Mounts</u>

The same comments apply to the design of mounts for these propellants as were made for oxygen-hydrogen system.

3. <u>Weights Analysis</u>

The same comments apply to the analysis for these propellants as were made for oxygen-hydrogen system. Figure 6 presents mass fractions for such a vehicle.

VII. MAN-RATING CONSIDERATIONS

Reliability is of prime importance in man rating a booster. The reliability objective is one in which the survivability is no less than exists for other risk occupations. The man rating of a booster includes provisions for a passenger escape system in the event of a failure in the booster that leads to a premature termination of the flight boost phase. In order to achieve high reliability in the booster, the system design should, basically, not be complex and have rigid quality control. Reliability of a successful flight is also formed by the number of normal events occurring within the booster during the flight and the number of normal events taking place in preparation of the flight.

In order to allow time for escape from the booster in the event of an improper operation of the booster, the monitoring of certain critical flight parameters must be accomplished.

A. STRUCTURAL

In order to man-rate the Titan structure it has been proposed that the current ultimate safety factor of 1.25 on limit load be increased to 1.4. In our opinion the safety factor should be maintained at the current value of 1.25 for the reasons below.

Once the structural loads have been adequately established, and one the missile structure's ability to withstand these loads has been verified by a suitable static test program, a factor of safety of 1.25 on limit load is sufficient to account for material and manufacturing variations and minor uncertainties in the structural loads. The Titan I flight test program has demonstrated the foregoing to be true. Increasing the safety factor from 1.25 to 1.4 will result in an overall increase in missile strength of 12 percent. This increase will then accommodate an additional uncertainty in structural loads of 12 percent. It also implies an increase of 6 to 12 percent in weight of load-carrying structure.

Man-rating the Titan I structure can be accomplished best by retaining the existing safety factor and expending maximum effort in the accurate determination of structural loads. This should be augmented by an extensive static test program for structural verification. As a final step, early unmanned flights of each payload configuration should be used to verify the predicted structural weight where necessary rather than arbitrarily increasing strength and weight at every station.

B. AERODYNAMIC

There are no specific differences in the aerodynamic aspects of a booster whether it is designed for manned or unmanned operation. The fin sizing criteria that the vehicle can never be statically unstable, that was applied to DynaSoar I, is no longer a requirement. The Mercury vehicle and the later DynaSoar configurations require that fins only provide enough stability so that the Tital vehicle can be controlled with reasonable engine gimbal angles and autopilot gains. A secondary requirement of the fins is that the angular acceleration caused by aerodynamic instability (in case of an engine failure at maximum dynamic pressure) be low enough so that the escape system can function before the missile is destroyed. In most cases, the latter requirement is less severe than the former.

In the areas of airloads and aerodynamic heating, the approach is the same for manned and unmanned vehicles. The heat protection provided for fire-in-the-hole staging is the only area that has been increased for the man-rated version. The heat protection in this area is increased to protect the first stage structure for two seconds at full thrust of the second stage engine to allow time for the escape system to function in case the stages do not separate after the ignition of the second stage engine.

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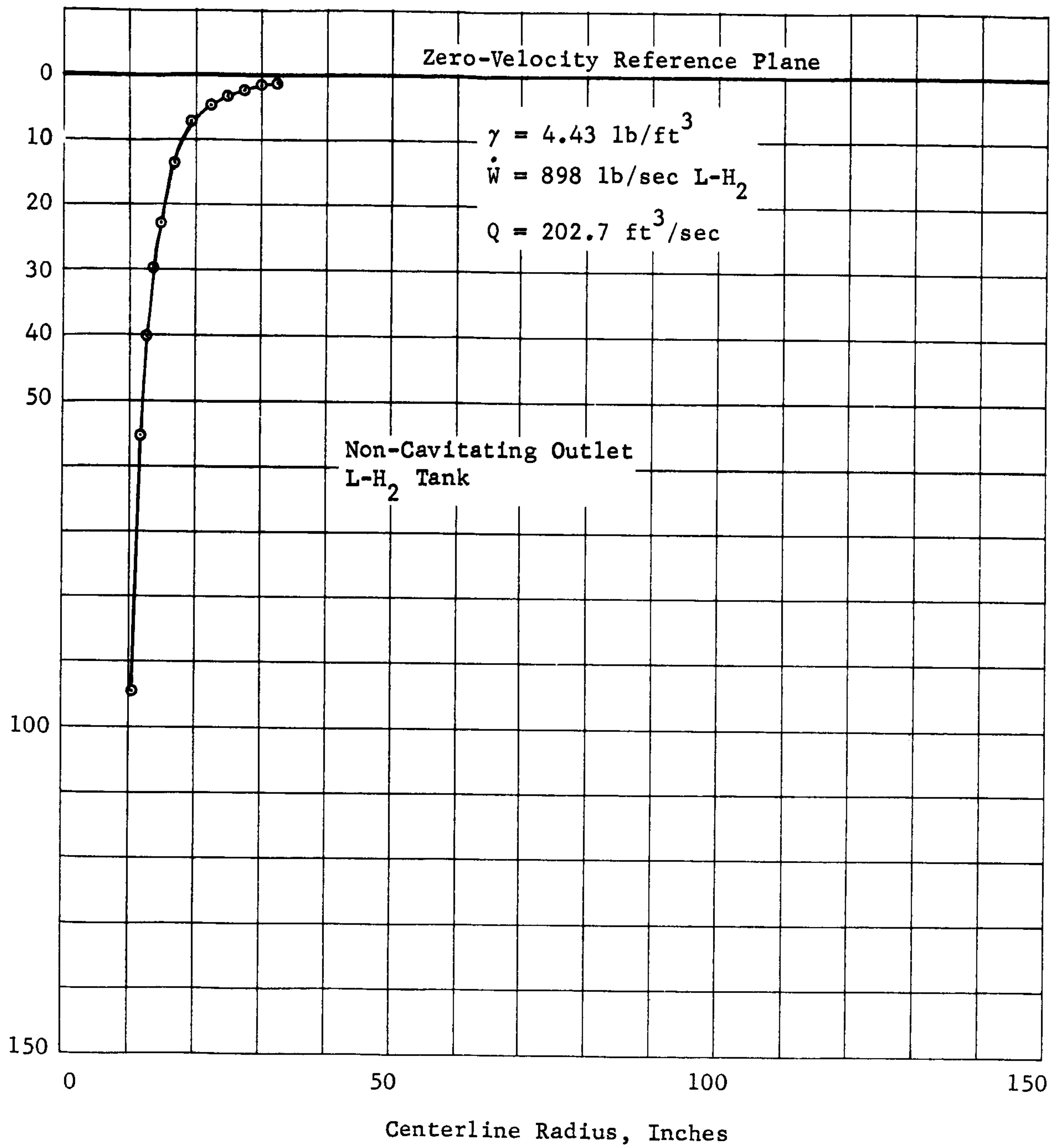
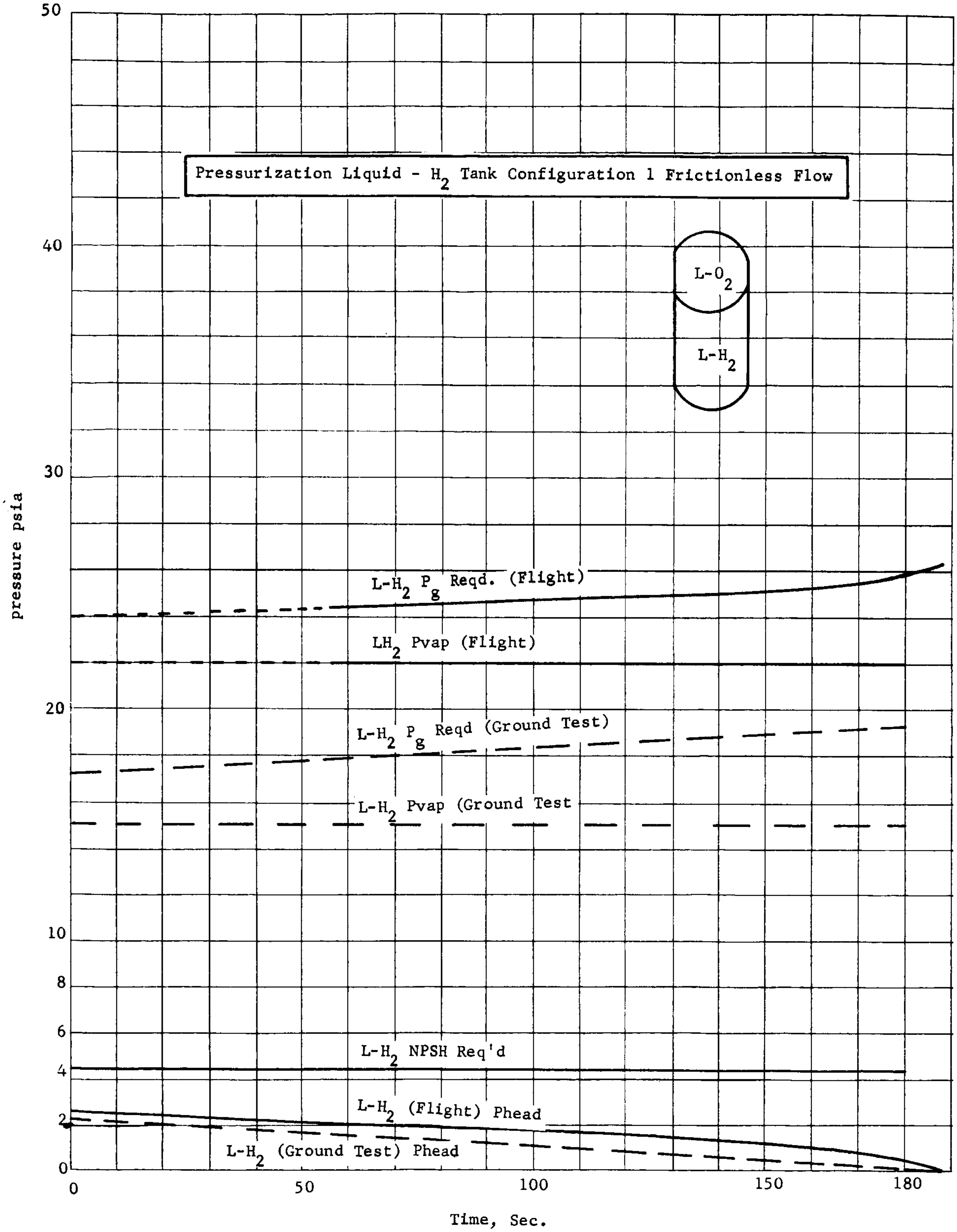

Zero-Velocity Reference Plane
γ = 4.43 lb/ft3
Ẇ = 898 lb/sec L-H2
Q = 202.7 ft3/sec
Non-Cavitating Outlet
L-H2 Tank
0
10
20
30
40
50
100
150
0
50
100
150
Centerline Radius, Inches

Figure 1


Pressurization Liquid - H_2 Tank Configuration 1 Frictionless Flow
L-O_2
L-H_2
L-H_2 P_g Reqd. (Flight)
LH_2 Pvap (Flight)
L-H_2 P_g Reqd (Ground Test)
L-H_2 Pvap (Ground Test
L-H_2 NPSH Req'd
L-H_2 (Flight) Phead
L-H_2 (Ground Test) Phead
pressure psia
50
40
30
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10
8
6
4
2
0
0
50
100
150
180
Time, Sec.

Figure 2

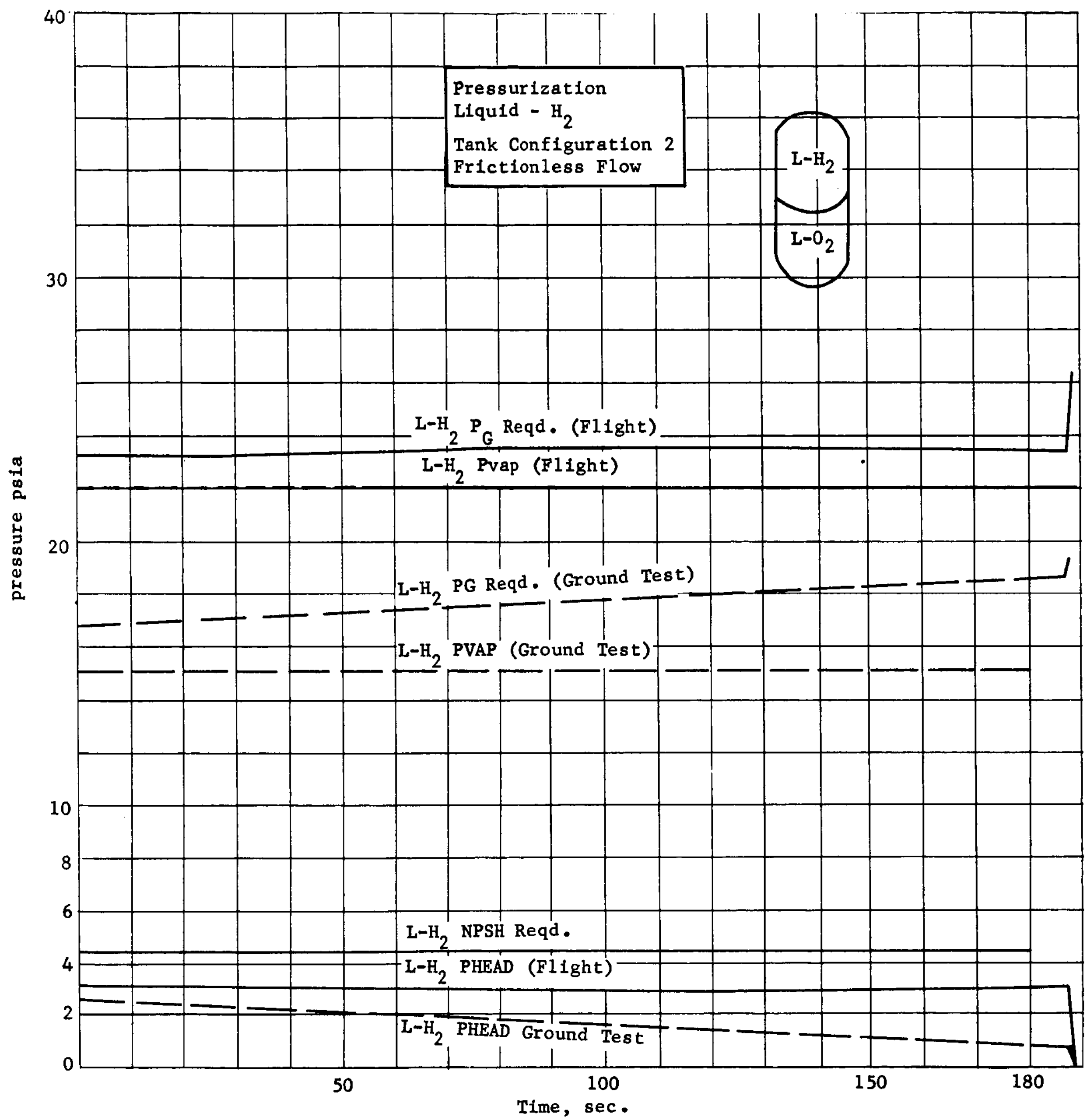
Pressurization
Liquid - H_2
Tank Configuration 2
Frictionless Flow
L-H_2
L-O_2
L-H_2 P_G Reqd. (Flight)
L-H_2 Pvap (Flight)
L-H_2 PG Reqd. (Ground Test)
L-H_2 PVAP (Ground Test)
L-H_2 NPSH Reqd.
L-H_2 PHEAD (Flight)
L-H_2 PHEAD Ground Test
pressure psia
40
30
20
10
8
6
4
2
0
50
100
150
180
Time, sec.

Figure 3

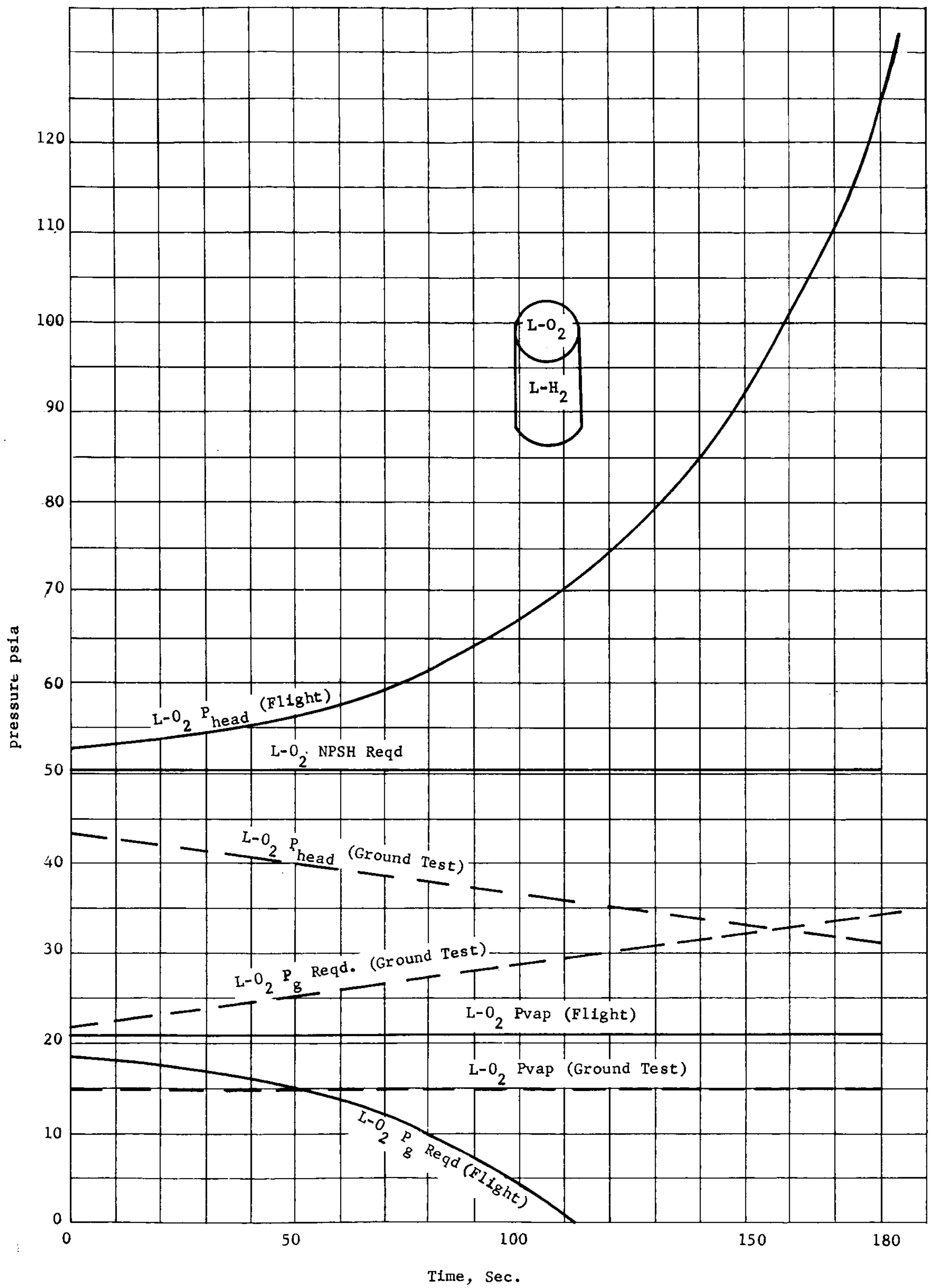

Figure 4 Pressurization Liquid - O_2 Tank Configuration 1 Frictionless Flow

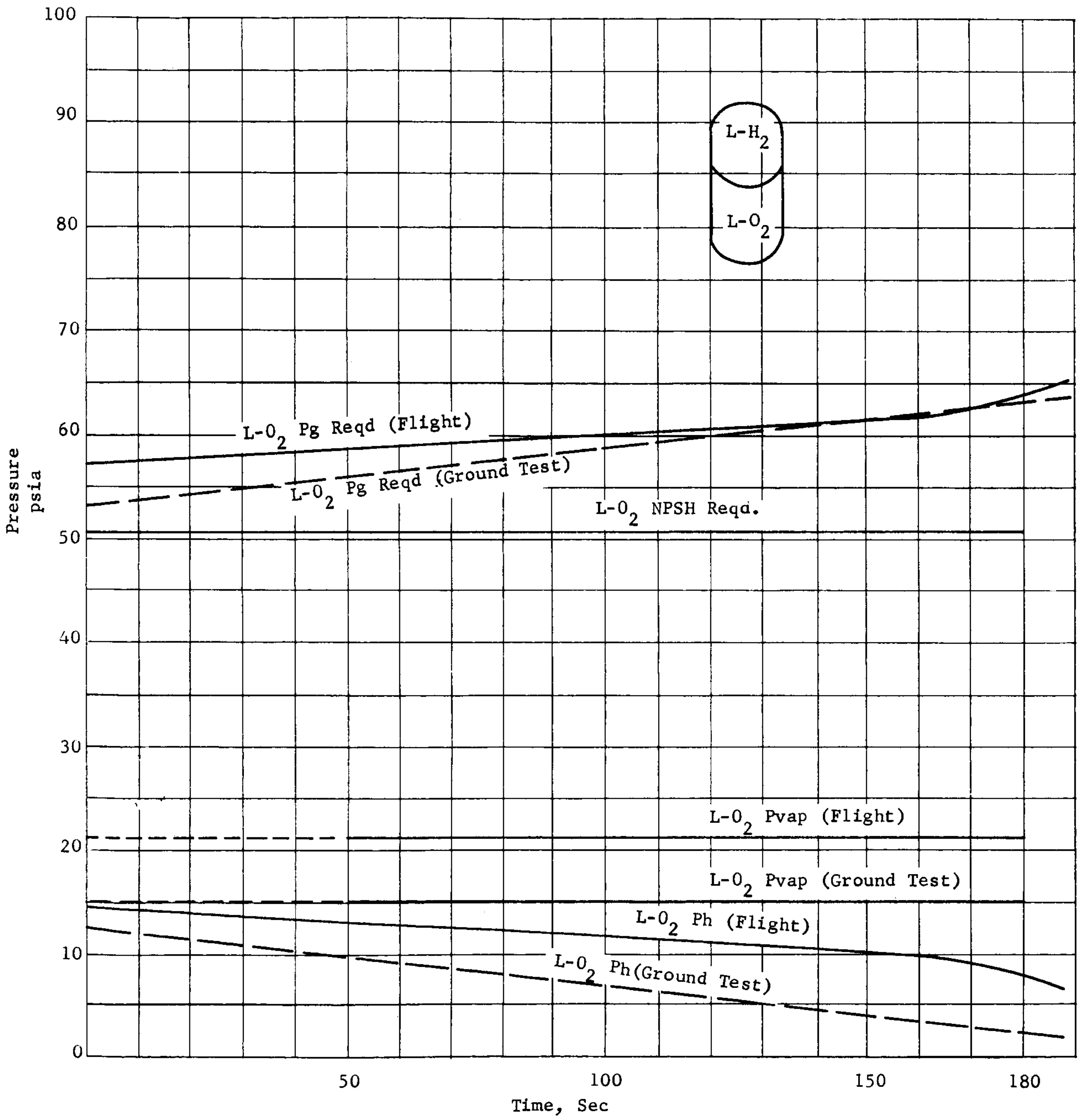

Figure 5 Pressurization Liquid - O_2 Tank Configuration 2 Frigtionless Flow

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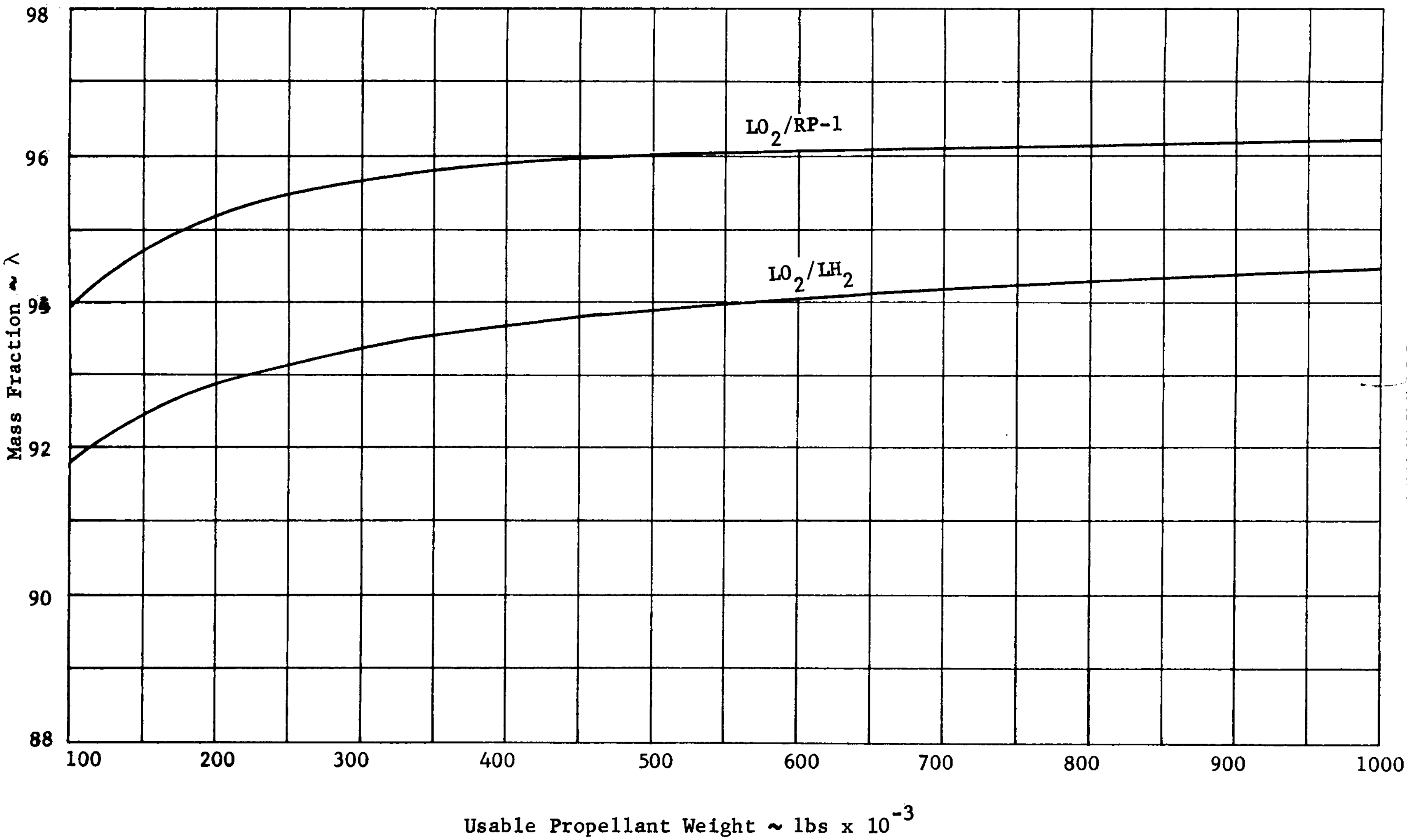

Figure 6 Mass Fraction vs Usable Propellant

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